N72-16720

NASA TECHNICAL MEMORANDUM

NASA TM X-2497

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DYNAMIC FLOW ANGLE MEASUREMENTS AT THE DIFFUSER EXIT OF A MACH 2.5 MIXED-COMPRESSION INLET

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1972

1. Report No. NASA TM X-2497	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle DYNAMIC FLOW ANGLE MEASUREMENTS AT THE DIFFUSER EXIT OF A MACH 2.5 MIXED-COMPRESSION INLET		5. Report Date February 1972
		6. Performing Organization Code
7. Author(s) William M. Prati and James E. Calogeras		8. Performing Organization Report No. E-6606
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		10. Work Unit No. 764-74
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		13. Type of Report and Period Covered Technical Memorandum
		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

An exploratory investigation was made to determine the magnitude of the dynamic flow angle at a single point in the diffuser exit of a Mach 2.50 inlet. Results indicate that the maximum flow angle variation was ±3° except when vortex generators were installed in the subsonic diffuser. The amplitude of these flow angle fluctuations was directly related to the amplitude of the total-pressure fluctuations at the probe survey point. Flow angle fluctuations as high as ±11° were recorded when vortex generators were installed near the probe survey point.

17. Key Words (Suggested by Author(s))		18. Distribution Statement	
Dynamic flow angle Inlet diffuser exit Flow direction probe, three-dimensional		Unclassified - unlimited	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 17	22. Price* $3.00

* For sale by the National Technical Information Service, Springfield, Virginia 22151

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SUMMARY

An exploratory investigation was made to determine the magnitude of the dynamic flow angle at a single point in the diffuser exit of a Mach 2.50 inlet. Results indicate that the maximum flow angle variation was $\pm 3^{\circ}$ except when vortex generators were installed in the subsonic diffuser. The amplitude of these flow angle fluctuations was directly related to the amplitude of the total-pressure fluctuations at the probe survey point. Flow angle fluctuations as high as $\pm 11^{\circ}$, recorded when vortex generators were installed near the probe survey point, appeared to be associated with the turbulent eddies in the wake of these generators.

INTRODUCTION

A test program utilizing an axisymmetric inlet designed for Mach 2.5 coupled to a J85-GE-13 turbojet engine was conducted in the 10- by 10-Foot Supersonic Wind Tunnel of the NASA Lewis Research Center. The primary objective of this investigation was to measure time-variant distortions produced in a supersonic inlet just prior to compressor stall. Results are presented in reference 1. A flow direction probe that measured time-variant flow angles was included in the program instrumentation. The probe was located at a stationary point in the subsonic diffuser of the inlet. This report delineates some of the results obtained from the probe.

Dynamic distortion is generally defined in terms of some magnitude of the total-pressure flucutations. However, the total-pressure instrumentation currently used to measure these fluctuations is relatively insensitive to variations in flow angle of as much as $\pm 20^{\circ}$ (ref. 2). And such flow angles could adversely affect the stability of the compressor. No dynamic flow angle data of this type are currently available in the literature. A possible reason for the lack of data is the difficulty in measuring these angles.

The measuring probe must be relatively small so that the velocity gradient across the probe tip is negligible. This isolates the flow angle variations from the effects of the velocity gradient. The probe must be capable of responding to frequencies that are consistent with the dynamics of the inlet. And despite its small size, the probe must be able to withstand the severe dynamic loads associated with inlet unstarts and compressor stalls (ref. 3).

A probe that met these requirements was designed, built, and installed in the inlet. Twenty different data points were reduced to define the magnitudes of the angle variations. Data that are presented include a trace of pitch and yaw angle variation with time as well as a table summarizing the peak-to-peak amplitudes of these flow angles for each of 20 different data points.

SYMBOLS

m/m_0 ratio of local mass flow rate to captured mass flow rate of stream tube (at free-stream Mach number) with cross-sectional area equal to projected cowl lip area

N engine speed, rpm

N^* rated engine speed, 16 500 rpm

P total pressure, N/m^2

$\mathscr{P}_1$, $\mathscr{P}_2$, $\mathscr{P}_3$, $\mathscr{P}_4$, $\mathscr{P}_5$ pressures in flow direction probe tubes (see fig. 6), N/m^2

ΔP fluctuating component of total pressure, N/m^2

p_{ref} stream static pressure at flow direction probe location, N/m^2

T total temperature, K

α flow angle in pitch plane at flow direction probe survey point, deg

β flow angle in yaw plane at flow direction probe survey point, deg

θ T/288.2 K

Subscripts:

L local conditions at flow direction probe survey point

o-p peak value of fluctuating flow angle

rms root-mean square

0 free-stream station

2 diffuser exit (compressor face) station

Superscript:

— area-weighted average

APPARATUS AND PROCEDURE

Inlet Details

The inlet used in this investigation was designed for Mach 2.50 and sized to match the airflow requirements of a J85-GE-13 turbojet engine. A photograph of the inlet terminated by a cold-pipe - choked-plug assembly is shown in figure 1 mounted in the Lewis 10- by 10-Foot Supersonic Wind Tunnel. While more complete inlet design and performance details are reported in reference 4, some of the more pertinent details are illustrated in figure 2.

The inlet was an axisymmetric mixed-compression type with 60 percent of the supersonic flow area contraction provided externally. Further isentropic compression resulted in a throat Mach number of 1.30. The compressor face station was segmented by three support struts which extended forward about half the length of the subsonic diffuser.

Engine Details

The General Electric J85-GE-13 is an afterburning turbojet possessing a high thrust-to-weight ratio. The engine consists of an eight-stage axial-flow compressor coupled directly to a two-stage turbine. It incorporates controlled compressor interstage bleed and variable inlet guide vanes, a through-flow annular combustor, and an afterburner (not used in this test) with a variable-area primary exhaust nozzle. The engine inlet diameter is 40.9 centimeters (16.1 in.).

The exhaust nozzle area was manually controlled for this investigation. The compressor was stalled by slowly closing the nozzle while maintaining a constant engine speed. In order to avoid exceeding the turbine temperature limit during this procedure, the first-stage turbine nozzle was approximately 14 percent smaller in area than the standard unit. At any point on the compressor map, then, the turbine was matched to the compressor at a lower turbine inlet temperature.

Instrumentation

Instrumentation at the compressor face station is presented in figure 3. This includes steady-state and dynamic pressure probes (discussed in more detail in ref. 1) as well as the dynamic flow direction probe. A photograph of the probe is presented in figure 4. This probe consisted of five tubes arranged in a cruciform. Each of these tubes, approximately 2.54 centimeters (1 in.) long, 0.1016 centimeter (0.040 in.) in outside diameter, and 0.0686 centimeter (0.027 in.) in inside diameter, terminated at the front face of a subminiature differential pressure transducer. The range of the differential transducers was 3.45 newtons per square centimeter (5 psi), and they were fitted with screens to protect the diaphragms from particle damage. A common reference pressure was supplied to each of these transducers from a cavity in the base of the probe housing. This cavity was bled to the local stream static pressure (p_{ref}). The position of the probe in the inlet is presented in figure 5.

Airflow orientation and tube pressure nomenclature are presented in figure 6 for the flow direction probe. The two outside tubes in both the pitch and the yaw planes were cut at 45° from the normal to obtain maximum sensitivity to flow angles in their respective planes. The center tube was left normal to the axial flow direction, and as such measured total pressure within 1/2 percent for flow angles as high as 20° (ref. 2).

Steady-state calibrations for the pitch and the yaw angles are presented in figure 7(a) and (b), respectively. These were made in a small tunnel facility at a free-stream Mach number of 0.4. A more detailed discussion of the steady-state calibration made with this type of probe is presented in reference 5. In figure 3 of reference 5, some data scatter is evident in the yaw angle calibration which is caused by the combined effects of Mach number, Reynolds number, and pitch angle dependency. This scatter, when applied to the calibration curves of figures 7(a) and (b), could amount to a maximum error of 10 percent in the pitch and yaw angles, respectively, that are presented in this report. A dynamic calibration of a two-dimensional flow direction probe is presented in reference 6.

Frequency response characteristics for one tube of the five-tube flow direction probe are presented in figure 8. Since all the tubes had practically identical pneumatic systems, their frequency characteristics were similar. The unfiltered characteristic showed a resonance at approximately 1200 hertz with an amplitude ratio of about 3.0. A second-order low-pass filter with a 100-hertz corner frequency was employed with each of these tubes for all the data reported. The filtered response is shown as the dashed curve of figure 8.

RESULTS AND DISCUSSION

Twenty data points are discussed in this report. They include data for three inlet configurations which differ in the use of subsonic diffuser section vortex generators. Inlet-engine operating condition variables include free-stream Mach number, inlet angle of attack, inlet bypass flow, and engine rotor speed. Inlet performance at Mach 2.50 and 0° angle of attack is presented in figure 9. Each datum point of this figure is numbered for subsequent reference.

Time histories of fluctuating pitch and yaw angles are presented in figure 10 for inlet-engine point 15. They were obtained by applying the steady-state calibrations of figure 7 to the time variant pressure data of the flow direction probe. This was accomplished through the use of an analog computer. These pressure data, recorded on FM-multiplexed tape, were filtered by the low-pass filter of figure 8. Only the fluctuations of the pitch and yaw angles were computed. Because of temperature effects, all the transducers in the flow direction probe experienced a zero shift during wind-tunnel operation. And the true steady-state values of tube pressures $\mathscr{P}_1$, $\mathscr{P}_2$, $\mathscr{P}_3$, and $\mathscr{P}_4$ could not be determined. Therefore, the pressure data from these transducers were filtered by high-pass filters to remove the erroneous steady-state outputs, and in this report, the pitch and yaw angles are assumed to fluctuate about 0°. This assumption has been validated in a subsequent investigation which showed that a similar inlet with the same subsonic diffuser streamlined the steady-state flow to within 3° of the axial direction. It was necessary, however, to compensate for the zero shift of the differential transducer sensing $\mathscr{P}_5$, since the actual differential pressure ($\mathscr{P}_5$ - p_{ref}) was required to maintain the same calibration sensitivity as shown by figure 7. By interpolating data from the steady-state total- and static-pressure instrumentation previously shown in figure 3, the proper bias was applied to the output of this transducer.

The time interval shown in figure 10 represents a period of about 200 milliseconds of a 5-second sample just prior to and including compressor stall. The large and rapid change in both the pitch and yaw angles at the end of this time interval is due to the hammer shock which results from a rapid expulsion of air following compressor stall. The stall zone in this compressor could have originated as much as 10 milliseconds before this time, as is shown by the cross-hatched region.

The traces shown are typical of the other 19 data points with the exception of the peak amplitude of the flow angle fluctuations. None of these traces appeared to be periodic, and the 200-millisecond interval shown in figure 10 was representative of the entire 5-second sample that was analyzed. Although their corresponding pressures were filtered above 100 hertz, higher frequency fluctuations in the computed pitch and yaw angles were evident. The peak-to-peak magnitudes of pitch and yaw angle, measured over the entire 5-second scan, are presented for each of the run numbers in table I.

Some mention should be made of the inaccuracies inherent in these flow angle measurements. As previously discussed, a maximum error of 10 percent in the computed flow angles could result from scatter in the steady-state calibrations. In addition, errors approximating 25 percent could result in the flow angle computation because of the uncertainty of the time variant local static pressure used in the definition of $\mathscr{P}_5$. This error was calculated by using the maximum amplitude of static-pressure fluctuations measured at the compressor face station during an investigation reported in reference 1. Transducer, recording, and interpolation inaccuracies could result in another 5 percent combined error. Therefore, the values of the flow angles presented in this report are felt to be within 40 percent of their true values.

Steady-state and dynamic pressure contours for point 15 are presented in figures 11(a) and (b), respectively. These contours were constructed from the measured pressures of the 30 steady-state and dynamic total-pressure probes located at the compressor face measuring station (fig. 3). For simplicity, the centerbody support struts which segment the inlet at this station were neglected in the construction of these contours. Each shaded region of the steady-state contour map represents a given range of total-pressure recovery, with the darkest region representing the lowest recovery. The boundary between any two shaded regions is a constant-pressure contour. The interpolated total pressures required for the $\mathscr{P}_5$ bias were obtained from contours of this type.

Each shaded region on the dynamic map represents a given range of total-pressure fluctuations. Here the darkest area represents the region of highest dynamic activity. To obtain the dynamic distortion contours, the output signal from each of the compressor face dynamic probes was filtered by a second-order low-pass filter with a 1000-hertz corner frequency and measured with a true rms meter. The rms amplitude of the fluctuating component of each total pressure was then divided by the steady-state compressor face average pressure, and the resulting value was used to construct the dynamic contour map.

A correlation between the peak amplitude of pitch and yaw angle and the rms amplitude of the fluctuating component of local total pressure divided by the local steady-state total pressure was found to exist. This correlation is presented in figure 12 for the inlet configuration having no vortex generators. With this configuration, both the pitch and yaw angle peak fluctuations increased linearly from about $1/2^{\circ}$ at a local dynamic distortion level of 0.01 to about 3° at a local dynamic distortion of 0.07. This low upper limit of flow angle was rather surprising in view of the results presented in reference 1. Those results showed that the high level of inlet dynamics measured for the same data point as point 9 in table I resulted in time-variant distortions of a type and amplitude completely different from that measured by steady-state instrumentation. Such startling changes in airflow distribution might be identified with a high degree of vorticity in the flow field. But the small flow angle fluctuations measured in this investigation are inconsistent with the theory that the large total-pressure fluctuations at the compressor face are caused by a vortical flow pattern.

For the configuration in which only centerbody vortex generators were used, an insufficient number of data points were reduced to make a positive correlation. But for the configuration in which both centerbody and cowl vortex generators were used, the reduced data were sufficient to show that no direct relation existed between the peak pitch and yaw angle fluctuations and the local dynamic distortion. The peak flow angle fluctuations for this configuration were generally higher than those of the other two configurations and in one case exceeded an amplitude of 10°. However, this is probably due to the proximity of the probe survey point to the turbulent eddies present in the wake of the vortex generators. This conclusion is supported by the observation that the local dynamic distortion levels are relatively low for these readings. In general for comparable inlet-engine operating conditions, that is, points 5 and 15, and 10 and 19, the addition of vortex generators significantly increased the flow angle variations while reducing the local dynamic distortion. This is consistent with the action of vortex generators.

A summary of the inlet-engine operating conditions and their corresponding steady-state total pressures, dynamic total pressures, and peak pitch and yaw angles is presented in table I. It is evident from these limited data that the total-pressure fluctuations produced in this inlet were predominantly due to fluctuations in the magnitude of the resultant velocity vector. This conclusion is apparent since the compressor face total-pressure probes used in this investigation were insensitive to flow angles up to 20°. Therefore, the high-response total-pressure probes used were adequate to describe the inlet dynamics.

SUMMARY OF RESULTS

An exploratory investigation was made to determine the magnitude of the dynamic flow angle at a single point in the diffuser exit of a Mach 2.50 inlet. Twenty data points were examined at various inlet-engine operating conditions. The following results were obtained:

1. Without vortex generators in the subsonic diffuser, maximum variation in flow angle was $\pm 3^{\circ}$. With vortex generators installed, the maximum value observed increased to $\pm 11^{\circ}$.

2. It appears that the total-pressure fluctuations produced in this inlet were predominantly due to fluctuations in the magnitude of the resultant velocity vector. Therefore, the high-response total-pressure probes used were adequate to describe the inlet dynamics.

3. The amplitude of the flow angle fluctuations was directly related to the amplitude of the total-pressure fluctuations at the probe survey point except when vortex gener-

ators were installed near this point. In general, it appeared that the larger flow angle fluctuations which were obtained when the vortex generators were installed may be associated with the proximity of the probe survey point to the turbulent eddies present in the wake of the vortex generators.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, November 5, 1971,
764-74.

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TABLE I. - SUMMARY OF DYNAMIC FLOW ANGLES AT VARIOUS INLET-ENGINE OPERATING CONDITIONS

Point	Free-stream Mach number, M_0	Inlet angle of attack, deg	Vortex generators			Engine speed, $N \times 100/N^* \sqrt{\theta}$, percent of rated	Mass flow ratio, m_2/m_0	Total-pressure recovery, $\overline{P_2}/P_0$	Local total-pressure recovery, $P_{2,L}/P_0$	Local dynamic distortion, $\Delta P_{L,rms}/P_{2,L}$	Pitch angle, α, deg	Yaw angle, β, deg
			None	Center-body	Cowl							
1	2.5	0	-	x	-	97.6	0.900	0.753	0.700	0.063	±7	±7
2	2.5	0	-	x	-	90.3	.874	.844	.859	.025	±4	±2
3	2.6	5	-	x	-	86.3	(a)	.857	.949	.014	±1	$\pm\frac{1}{2}$
4	2.5	0	x	-	-	86.8	.870	.918	.950	.010	±1	$\pm\frac{1}{4}$
5	2.5	0	x	-	-	92.8	.890	.833	.900	.044	$\pm 2\frac{1}{2}$	±2
6	2.5	0	x	-	-	95.6	.891	.799	.880	.064	±3	±2
7	2.5	0	x	-	-	98.8	.892	.761	.849	.072	±3	±3
8	2.5	0	x	-	-	86.7	.816	.861	.920	.028	$\pm 1\frac{1}{2}$	±2
9	2.5	0	x	-	-	86.9	.746	.788	.849	.074	±3	±3
10	2.5	0	x	-	-	90.2	.833	.829	.899	.042	$\pm 1\frac{1}{2}$	±2
11	2.5	0	x	-	-	92.3	.855	.800	.849	.061	$\pm 2\frac{1}{2}$	$\pm 2\frac{1}{2}$
12	2.6	5	x	-	-	92.9	(a)	.771	.949	.024	$\pm\frac{1}{2}$	±1
13	2.6	5	x	-	-	86.9	(a)	.844	.948	.009	$\pm\frac{1}{2}$	$\pm\frac{1}{4}$
14	2.6	5	x	-	-	86.9	(a)	.806	.949	.007	$\pm\frac{1}{2}$	±1
15	2.5	0	-	x	x	90.4	.891	.855	.824	.010	±6	±6
16	2.5	0	-	x	x	96.0	.900	.768	.774	.020	$\pm 1\frac{1}{2}$	±1
17	2.5	0	-	x	x	87.0	.814	.838	.819	(b)	±7	±7
18	2.5	0	-	x	x	87.3	.717	.738	.734	.020	±10	±11
19	2.5	0	-	x	x	89.9	.829	.795	.779	.015	±7	±7
20	2.5	0	-	x	x	90.1	.799	.766	.760	.020	$\pm 3\frac{1}{2}$	±3

[a]Inconclusive data.
[b]No data.

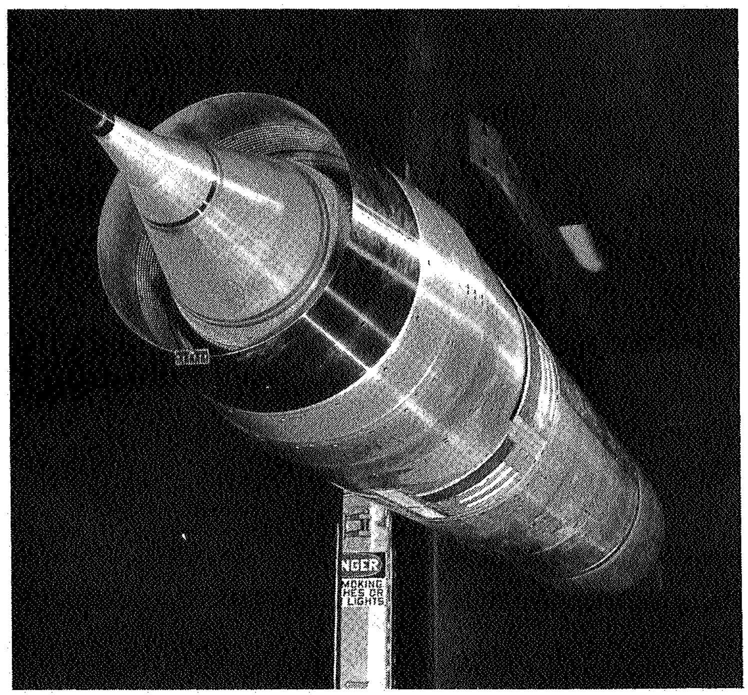


Figure 1. - Inlet - cold-pipe installation in 10- by 10-foot supersonic wind tunnel.

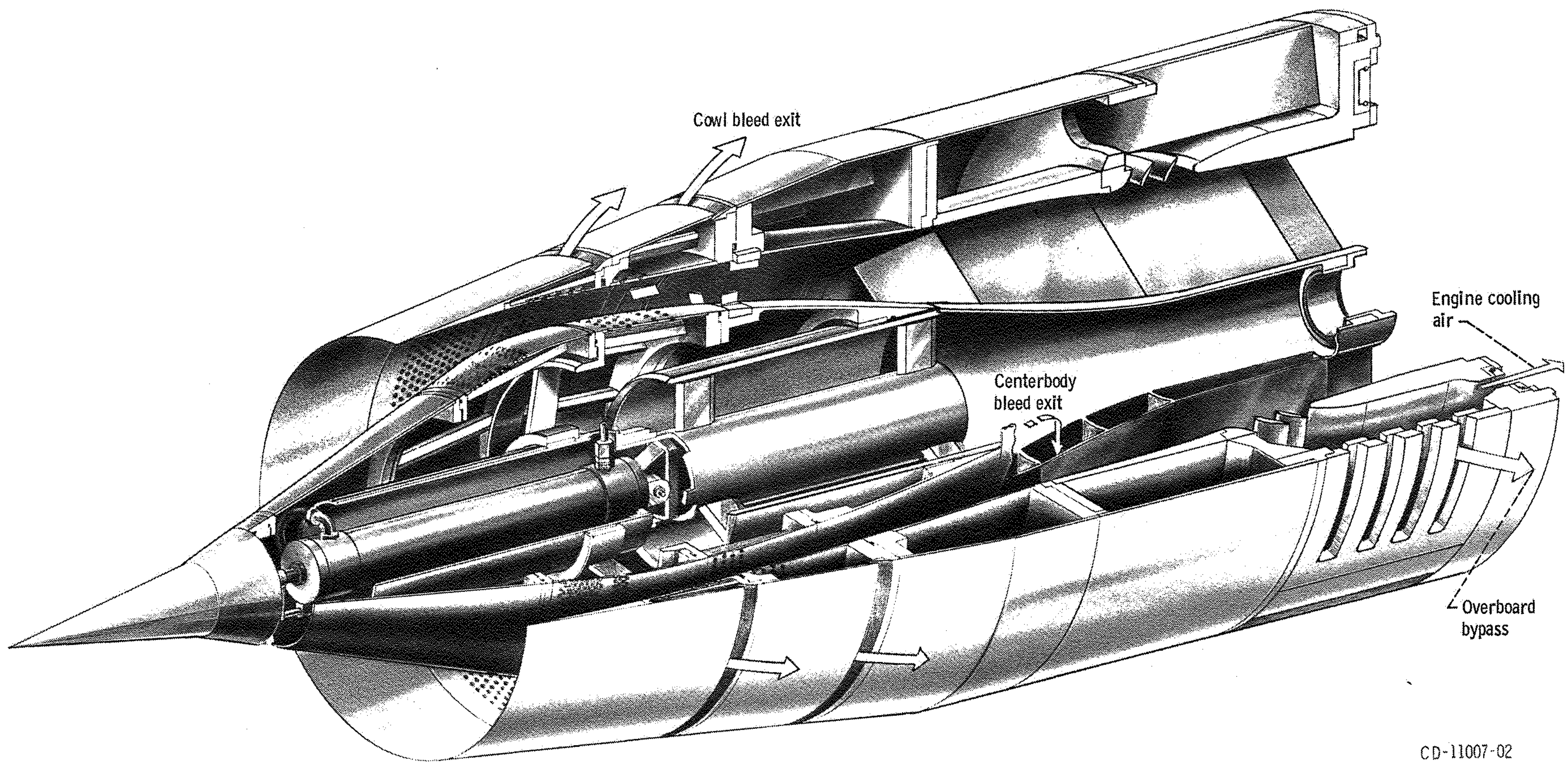


Figure 2. - Inlet details.

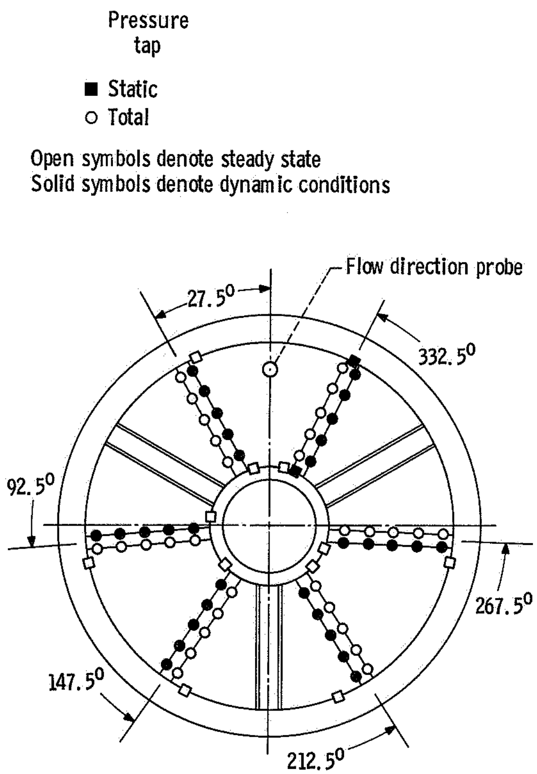


Figure 3. - Compressor face instrumentation. View looking downstream.

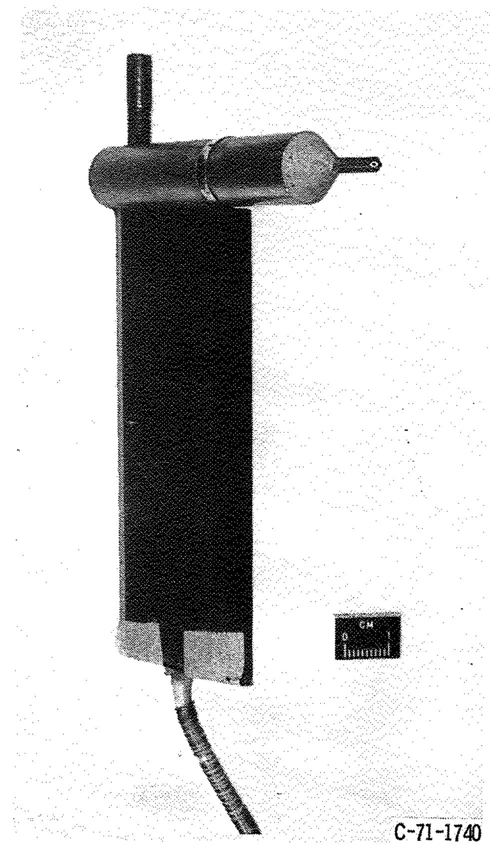


Figure 4. - Three-dimensional flow direction probe.

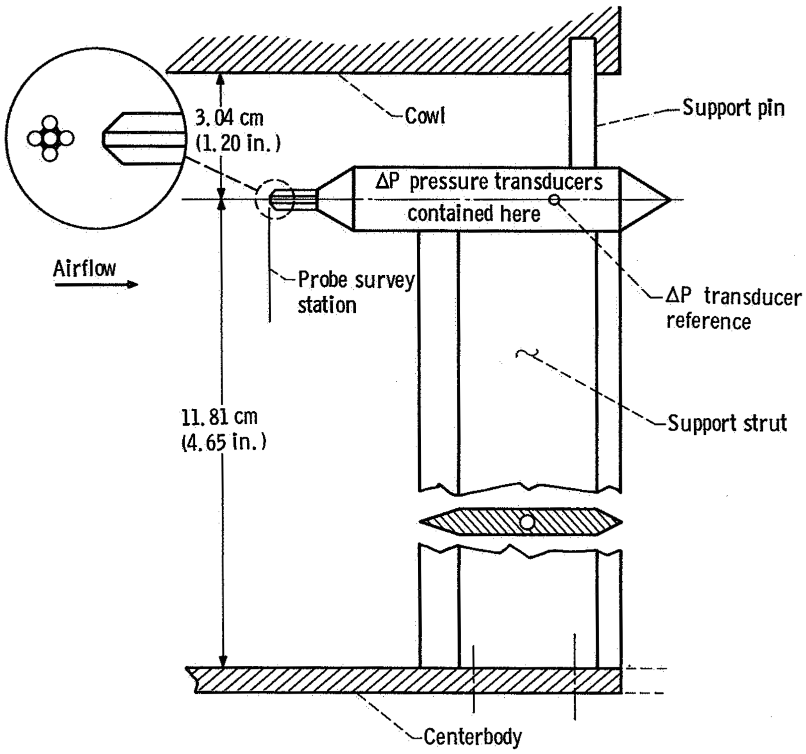


Figure 5. - Three-dimensional flow direction probe position. Probe survey station is located approximately 12.7 centimeters (5 in.) upstream of engine inlet guide vanes; probe tip is located angularly at 12 o'clock position.

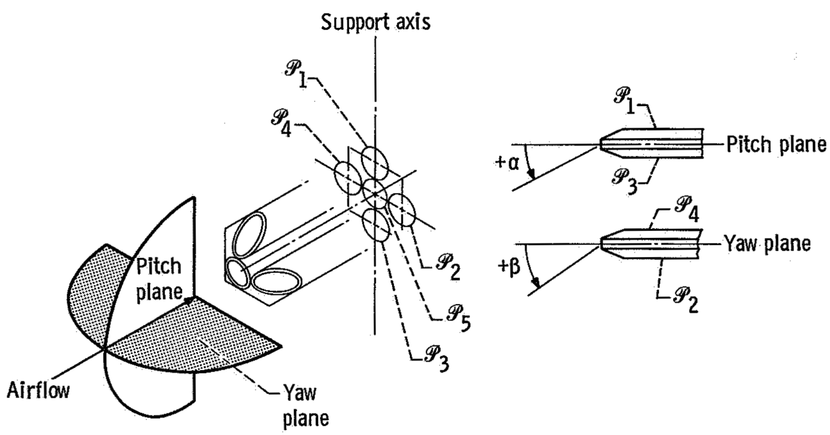
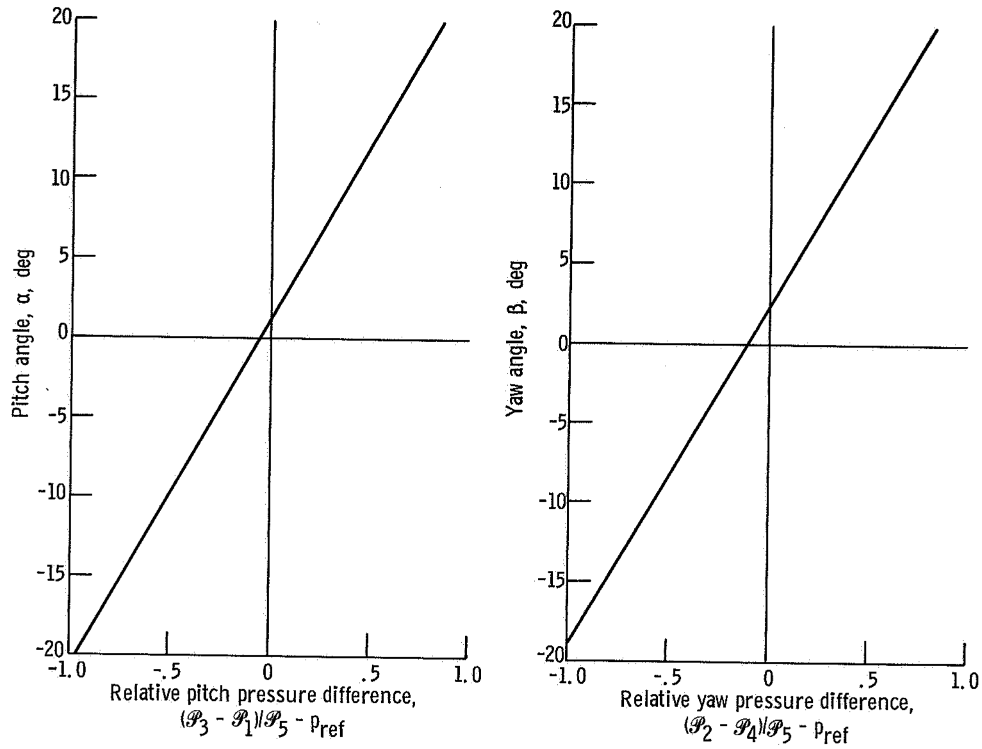


Figure 6. - Probe flow orientation and tube pressure nomenclature.



(a) Pitch. (b) Yaw.

Figure 7. - Probe steady-state angle calibrations. Mach number, 0.4.

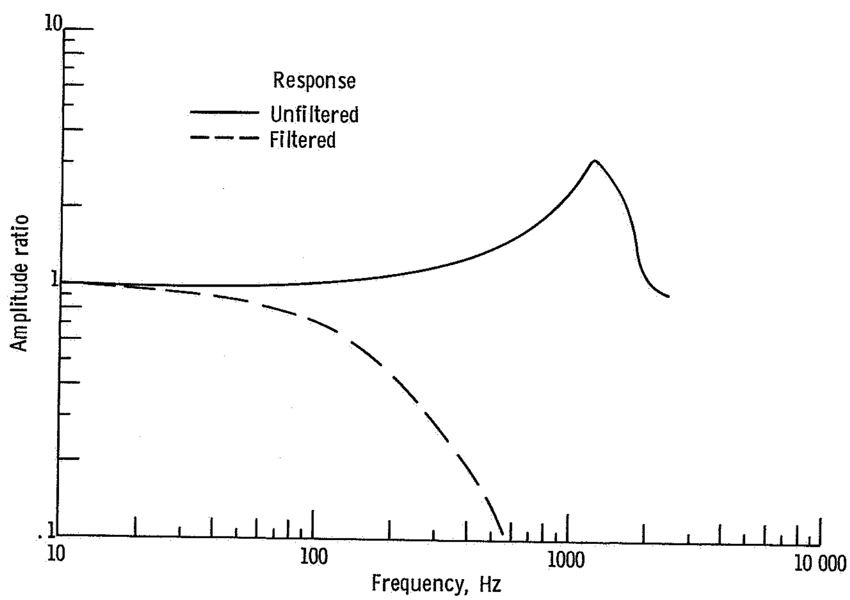


Figure 8. - Frequency response characteristics of typical tube of probe configuration.

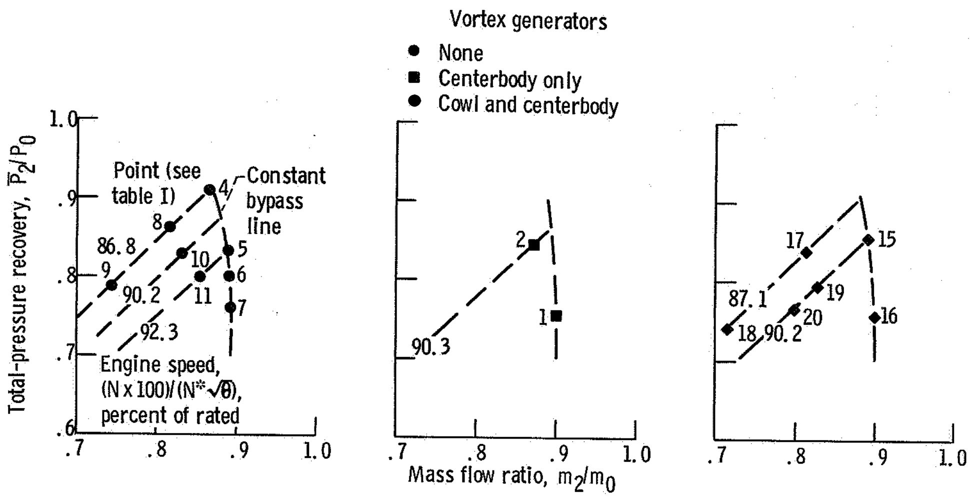


Figure 9. - Inlet performance at Mach number of 2.50 and angle of attack of 0°.

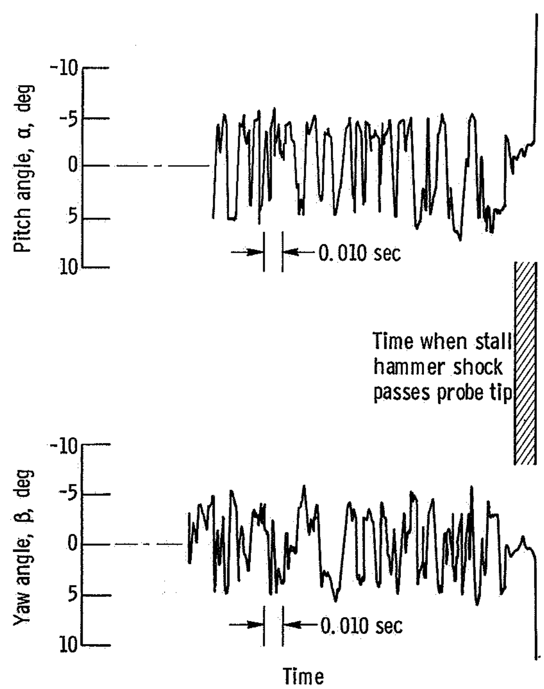
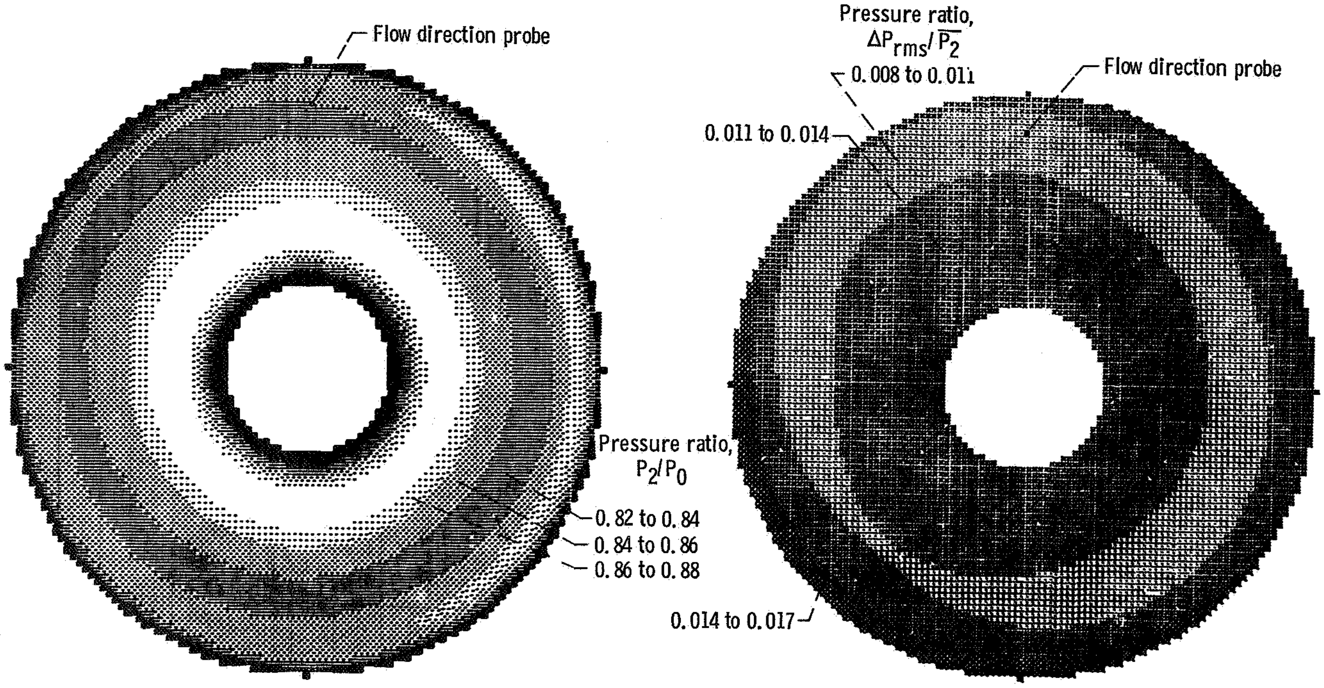


Figure 10. - Pitch and yaw angle time history for point 15.



(a) Steady state.

(b) Dynamic conditions.

Figure 11. - Compressor face distortion contour for point 15.

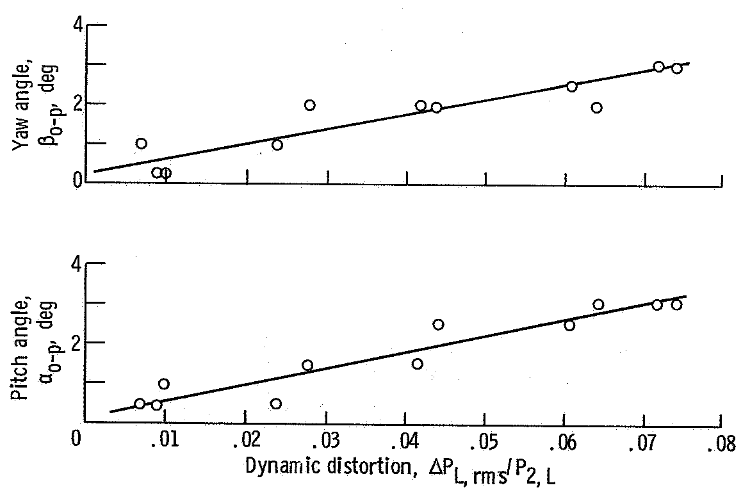


Figure 12. - Pitch and yaw angle as function of dynamic distortion at probe location. No vortex generators.

NASA-Langley, 1972 —— 28 E-6606